111-18
114301
p. 38

NASA Technical Memorandum 107577

A Conceptual Design of a Large Aperture Microwave Radiometer Geostationary Platform

Paul A. Garn
James L. Garrison
Rachel Jasinski

June 1992

National Aeronautics and Space Administration
Langley Research Center
Hampton, Virginia 23665-5225

TABLE OF CONTENTS

LIST OF SYMBOLS AND ACRONYMS

BOL	Beginning Of Life
C&DH	Control and Data Handling
CMG	Control Moment Gyroscope
DOD	Depth of Discharge
e/cm^2	Electrons per square centimeter
EOL	End Of Life
F	Fluence (e/cm^2)
GCPS	Global Change Process Study
GEO	Geostationary Orbit
g_o	Earth's gravitational value at sea level ($9.81\ m/s^2$)
Hr	Hour
Isp	Specific Impulse (s)
kg	Kilogram
LAMR	Large Aperture Microwave Radiometer
LEO	Low Earth Orbit
km	Kilometer
KW	Kilowatt
m	meter
M	Mass (kg)
MACS	Modular Attitude Control System
MeV	Mega-Electron Volts
mil	Thousandths of an inch
MLI	Multi-Layer Insulation
MMS	Multimission Modular Spacecraft
$M_{P/S}$	Mass of Propulsion System for Orbit Transfer
N	Newton (Kgm/s^2)
NASA	National Aeronautics and Space Administration
n/p	Solar Cell Junction Type
OTV	Orbital Transfer Vehicle
P_e	Electric Power
P_o	Initial Power
POST	Program to Optimize Simulated Trajectories
P_T	Electric Power Delivered to Ion Propulsion System
PV	Photo-Voltaic
RMS	Root Mean Squared
RTG	Radio-Isotope Thermal Generator
SP-100	Space Power - 100 KW of Nuclear Power
t	Time
T	Thrust
TEA	Torque Equilibrium Attitude
UARS	Upper Atmospheric Research Satellite
V	Velocity
V_o	Initial Velocity
W	Watt

LIST OF FIGURES

LIST OF TABLES

1.0 INTRODUCTION

An understanding of the physical and chemical processes that govern the global environment has been set as a priority by numerous national and international committees. In response to reports, such as the Augustine report (reference 1) and the Ride report (reference 2), NASA is making a significant effort to develop technologies that enhance the Global Change Process Studies (GCPS). One of the best techniques for studying the Earth's atmospheric processes is microwave remote sensing. Remote measurements must be made from space to understand both regional effects and global processes such as ozone depletion, atmospheric-ocean heat transport, and mankind's role in the build-up of "green house" gases.

The application of space-borne microwave remote sensing technology will provide the needed information on rainfall, rainfall rates, windspeeds, surface temperature, sea level variations, and other surface and atmospheric phenomena. The location (orbit altitude and inclination) of the remote sensing device, the required detail of the measurement (resolution), and the required interval at which the measurement is taken (temporal coverage) are major factors to be considered in the conceptual design of space-based remote sensing systems.

An advisory group to NASA, the Large Space Antenna Science Benefits Panel has stated that most of the research related to atmospheric and surface processes could be accomplished from geostationary orbit (reference 3). The panel also concluded that a minimum antenna aperture of 25 m was required in this orbit to provide the necessary resolution to satisfy the science requirements. However, both the sensor and the platform requirements for an antenna of this size, with large focal lengths and strict platform stability requirements, are beyond the current state-of-the art.

The purpose of this study is three fold: (1) integrate on-going Office of Aeronautics and Space Technology's (OAST) microwave antenna sensor activity into a flight mission; (2) establish a viable conceptual design for a Large Aperture Microwave Radiometer (LAMR) and supporting platform combination and (3) identify enabling technologies in the area of space platforms.

2.0 ASSUMPTIONS AND SCOPE

The conceptual design of the LAMR platform presented in this paper is based on several assumptions and restrictions. The LAMR platform design study is limited to those technologies that could be flight ready around the year 2002. The LAMR sensor design assumes the use of a Foldes "type six" antenna as a baseline, although the platform design presented could be easily adapted to a Foldes "type two" antenna, both of which were recommended by the Science Benefits Panel (Reference 3). The LAMR sensor design in this study is limited to mechanical characteristics of the sensor, therefore no electromagnetic design was done on the LAMR sensor or any of its components.

Another area for advancement is in areal density, the ratio of total mass of the reflector to the usable reflective surface area. The current areal density for large antenna is 5 kg/m^2. In this study, an areal density of 3 kg/m^2 is assumed for the large antenna, since current technology studies are predicting a trend toward such areal densities (reference 4).

Two antenna construction concepts (erectable and deployable) were considered for the conceptual design. Both antenna types are to be constructed in Low Earth Orbit (LEO), at an altitude of 500 km with an inclination of 28.5 deg. The launch system assumed for this study is a combination of the Space Shuttle and, if necessary, an additional expendable launch vehicle. Upon completion of the construction, the LAMR platform is boosted from LEO to Geostationary Orbit (GEO). A maximum acceleration limit of 0.1 g_o is imposed by the structural design. The transfer trip time is limited to one year.

This study is restricted to a conceptual design of a platform to support a microwave sensor in a geostationary orbit. Since some of the recommendations for systems and their components are beyond the current state-of-the art, the level of detail for the study is generally restricted to the systems level, where applicable individual components are selected.

3.0 ANTENNA SENSOR CONCEPT DEFINITION

The principal element of the LAMR platform is the large antenna that constitutes the microwave sensor. The requirements for the microwave sensor are given in Table 1 (reference 3).

Orbital Altitude (km)	35,863 (GEO)
Orbital Inclination (deg)	0
Life Time (years)	5
Minimum Aperture (m)	25
Resolution (km)	5-30
Frequency (GHz)	19-60
RMS surface accuracy (m)	1×10^{-4}
Pointing Accuracy (deg)	0.05

Table 1 Mission Sensor Requirements

Two general categories of large antennas, based on the construction method, deployable and erectable, are considered for the advanced microwave radiometer. Both mesh and solid reflector surfaces are available for these antennas. A solid reflector surface is required to accommodate the mission sensor requirements (reference 5).

A cursory study of a deployable microwave radiometer with a solid reflector surface was conducted by Astro Aerospace Corporation (reference 6). In this study, a 28 m primary reflector (with a 25 m diameter aperture), 12 m diameter secondary reflector, support mast, and feed array (Figure 1) were packaged to use the entire Titan IV dynamic envelope (Figure 2). Although the Astro Aerospace study shows that the sensor could be delivered to orbit in one flight, it did not include important subsystems like the spacecraft bus, sensor control electronics, propellent, etc. This type of reflector design would require two launch vehicle flights and some on-orbit assembly.

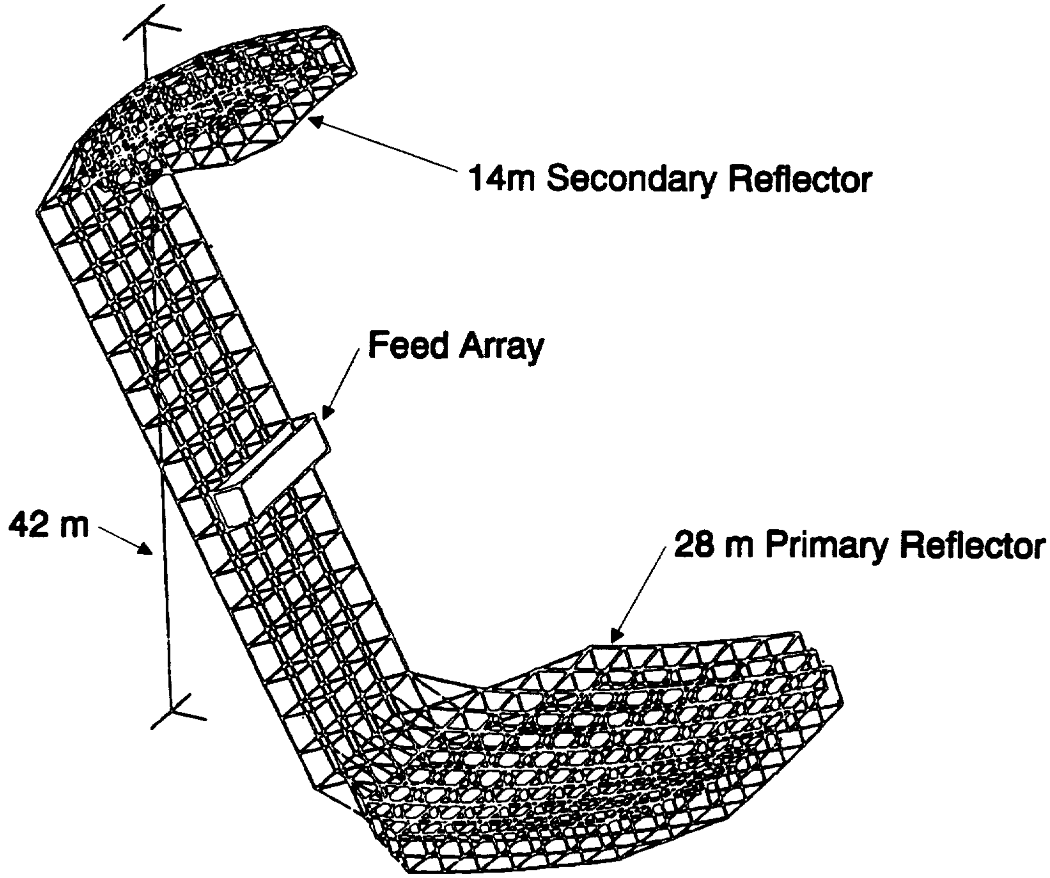

Figure 1 28 m Deployable Radiometer Concept

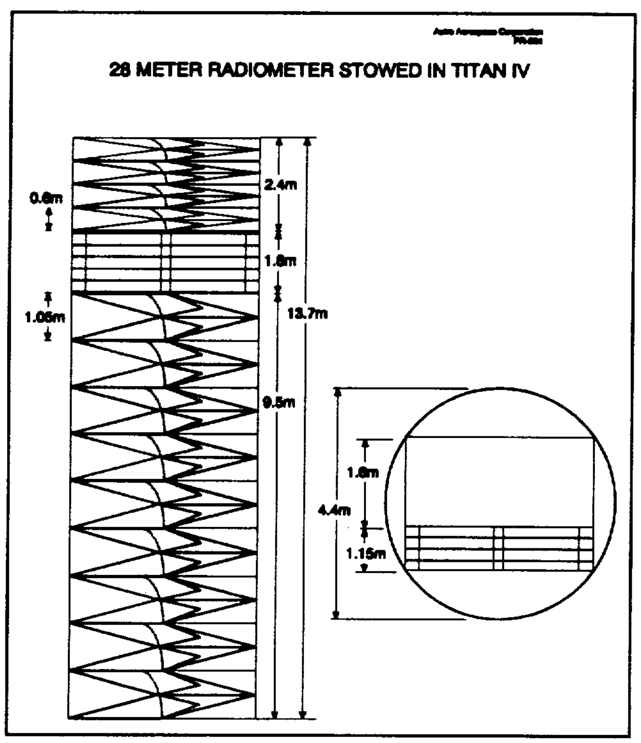

Figure 2 28 m Deployable Radiometer Concept Stowed in Titan Shroud

A recent study by LaRC's Structural Concepts Branch (Reference 7) has shown assembly scenarios for erectable truss structures in space. Plans to construct a 12 m reflector and to conduct neutral buoyancy tests on the strong back structure and parts of the reflector surface are included; however, the assembly of the total sensor/platform is not addressed.

For the purposes of this study it is assumed that reflectors as large as 28 m can be assembled on-orbit, with appropriate technology development. The proposed LAMR sensor configuration is show in Figure 3.

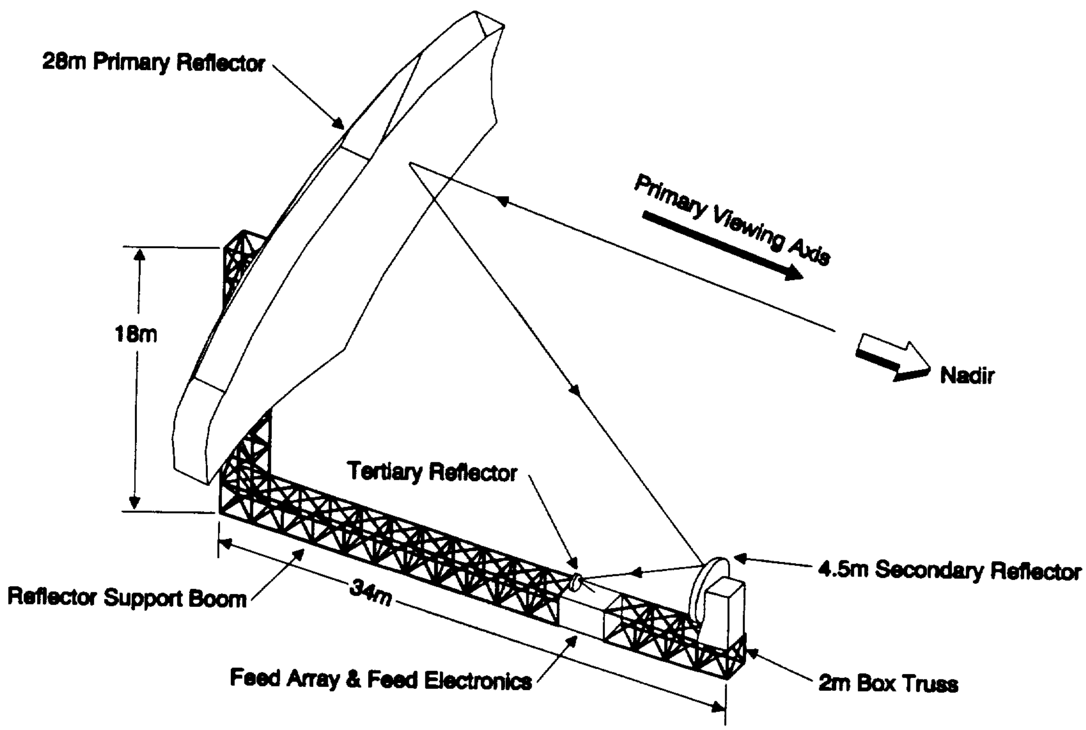

Figure 3 Large Aperture Microwave Radiometer Configuration

The LAMR sensor consists of three reflector surfaces, a support boom, feed assembly, and additional structure for the secondary reflector. In Figure 3, the support boom is aligned along the primary viewing axis, which is pointed toward the Earth during operation of the sensor. The primary reflector is an offset 28-m diameter parabolic reflector with a 25-m diameter aperture (a 25-m projected circle inscribed in a 28-m hexagon). Radiated energy from the Earth, in the microwave length region is reflected by the primary reflector to the secondary reflector. The secondary reflector is a 4.5-m diameter solid reflector similar to the one described in reference 8. The microwave energy passes from the secondary reflector to the tertiary reflector, which is mounted on the feed array assembly. Finally, the microwave energy is reflected from the

tertiary reflector to the feed array inside the feed assembly housing. The entire sensor is supported by a 2-m box truss. After assembly of the primary reflector and support boom, both components are covered with Multi-Layer-Insulation (MLI) to provide thermal stability for the reflector and boom portion of the LAMR instrument. The integration of the cabling required for power and data is included in the assembly of these components.

The erectable sensor concept was selected for this study, however the comments are applicable to either sensor concept.

4.0 LAMR PLATFORM CONCEPT DEFINITION

Having defined the erectable LAMR sensor, the supporting platform is examined. In this study, six platform systems are assessed: platform power, on-orbit propulsion, attitude control, communications and control, thermal control, and structures. These systems are assessed for both the transfer trajectory, from LEO to GEO, and on-orbit operations. The overall LAMR platform is shown in Figure 4.

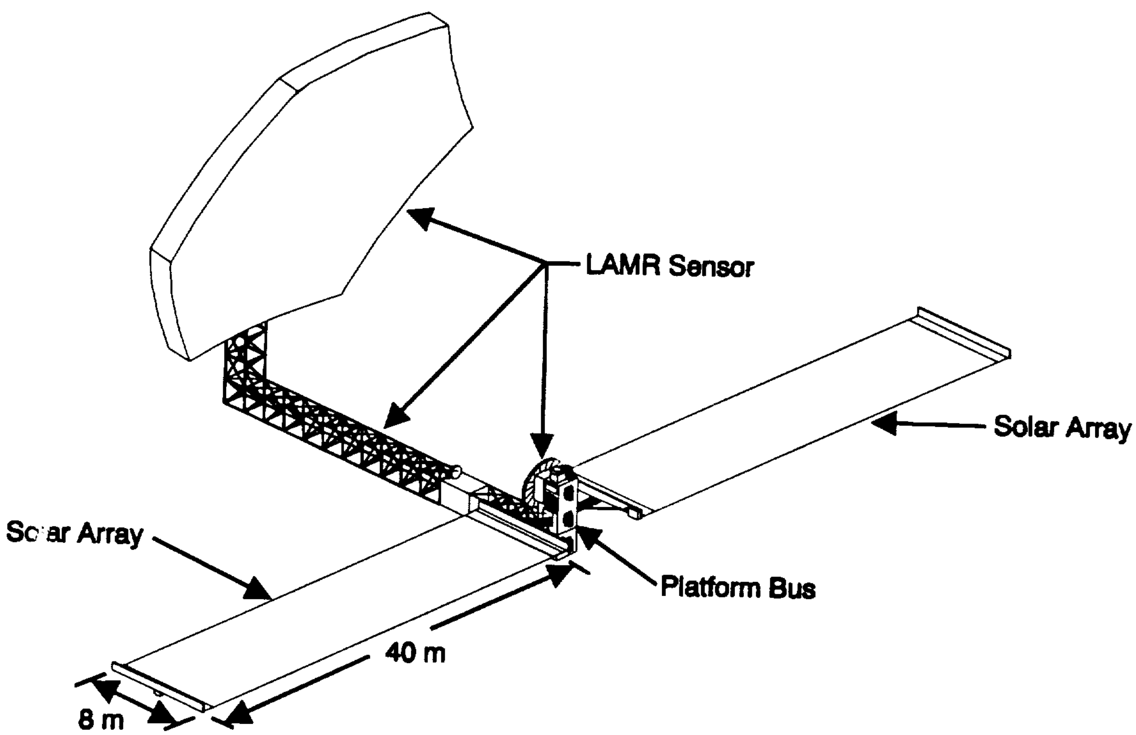

Figure 4 Integrated LAMR Platform

A summary of LAMR platform characteristics is detailed in Table 2. The mass characteristics of each system are shown in Table 3.

Instruments:	LAMR Sensor, GOES Imager
Platform Power:	Indium Phosphide Solar Cells, NiH_2
Platform Propulsion:	Ion Electric Thrusters, Fuel Xenon
Attitude Control:	Three-Axis Stabilized using CMGs
Thermal Control:	Passive, Except Heaters for Batteries

Table 2 LAMR Platform Characteristics

SYSTEM	(kg)	(kg)
Sensor		
LAMR Sensor	2949.0	
GOES Imager	118.0	
Mass Margin (10%)	306.7	
Total Sensor Mass	**3373.3**	
Platform		
Platform Power		237.9
Platform Propulsion		223.6
Attitude Control		510.0
TT&C		130.0
Thermal Control		74.0
Structures		430.0
Miscellaneous		100.0
Mass Margin (10%)		170.5
Total Platform Mass		**1876.1**
Total LAMR Platform Mass		**5249.4**

Table 3 LAMR Platform Systems Mass Summary

4.1 PLATFORM POWER SYSTEM

The power system is composed of two subsystems: a non-regulated 28 Volt power system that drives the orbital transfer electric propulsion system (section 5.2) and a fully regulated 28 Volt power system that provides LAMR platform power. The mass of the first system is included in the propulsion system mass. The mass of the second system is comprised of the mass of additional propulsion solar arrays, the platform electrical power control electronics, and the platform batteries.

The platform power system, including the batteries and solar arrays, is designed to take into account the power required during both operational periods of the platform, the transfer from LEO to GEO and on-orbit operations at GEO. Each of these operating conditions poses a distinct challenge to the power system. First, during the transfer phase, the platform bus has a continuous power requirement of 1.0 KW. The power system must be sized to provide the 1.0 KW of continuous power as well as allow for the 26 percent power system loss resulting from radiation degradation during transfer (see section 5.2.3). Therefore, the platform power system is designed to produce 1.57 KW of power at Beginning Of Life (BOL). Second, during the operational phase at GEO, the spacecraft sees its largest eclipse period (1.2 hours), giving the largest Depth of Discharge (DOD) on the batteries. The spacecraft bus requires 1.5 KW of power

while an additional 0.24 KW is needed to charge the batteries. Sizing to include the power system loss, the bus requirement and the power to charge the batteries, the power system is designed to produce 2.6 KW of power BOL. The platform uses NiH_2 batteries designed for 128.6 Ahr to accommodate both of these conditions. Using a conservative figure of 55 Whr/kg and a DOD of 50 percent, the battery is sized at 65.5 kg. The solar array mass for the platform power system is estimated to be 13.1 kg (using 120 W/kg), and the power system control electronics are estimated to be 95.3 kg (reference 9). The above power system does not include the propulsion power system (section 5.2.3).

The same arrays are used for both operational modes. During the transfer, the entire solar arrays, including the propulsion system solar arrays, are deployed to maximize the power output. Once in GEO, the arrays are mostly retracted to reduce drag and power output. Once retracted, only 32 m^2 are exposed to sunlight. This is the area necessary to produce the required 2.6 KW BOL power. This area includes the 26 percent power loss sustained during transfer from exposure to the radiation environment in the Van Allen belts.

4.2 PLATFORM PROPULSION SYSTEM

The electrical propulsion system selected for orbital transfer (section 5.2.3) is also used for GEO on orbit operations. A summary of the platform operational propulsion system characteristics are shown in Table 4. The selected electrical propulsion system uses xenon ion thrusters with an Isp of 4200-sec for GEO orbit maintenance and Control Moment Gyroscope (CMG) desaturation. A typical thruster configuration is displayed in Figure 5.

Latitude Tolerance (deg)	0.1
Longitude Tolerance (deg)	0.1
Delta V required (m/s)	523.6
Operational Mass (kg)	12,491
Orbit Maint. Prop. Mass (kg)	157.7
CMG Desat. Prop. Mass (kg)	65.6
Life Time (years)	7

Table 4 Platform Propulsion Characteristics

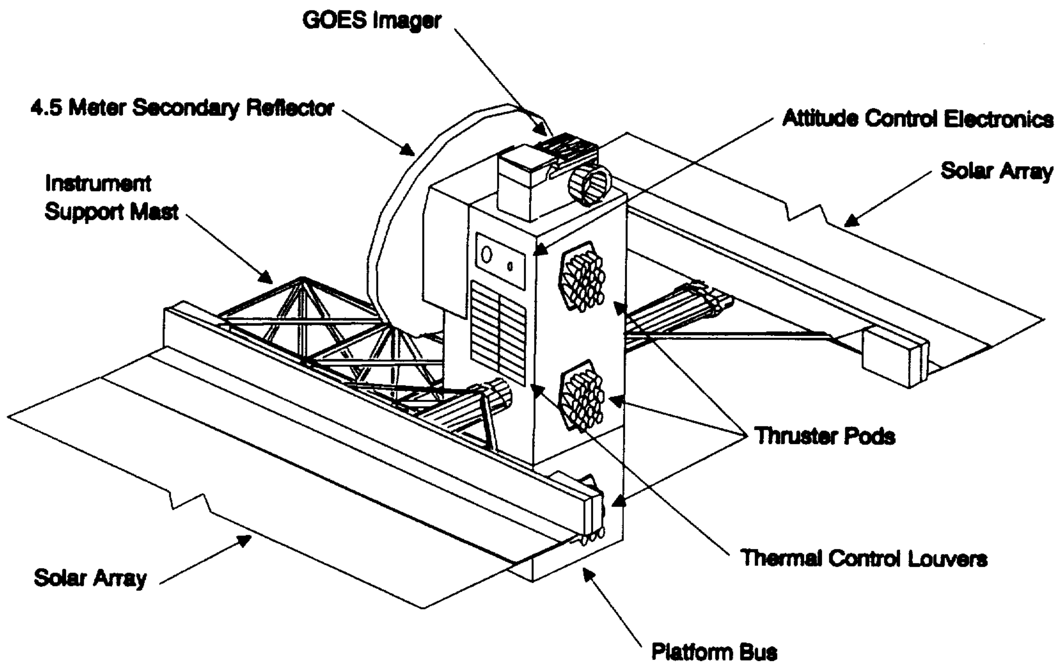

Figure 5 LAMR Platform Bus Detail

4.3 ATTITUDE CONTROL SYSTEM

The attitude control system uses the Fairchild Multimission Modular Spacecraft's (MMS) Modular Attitude Control System (MACS) module. The details of the MACS are in Table 5. The MACS is supplemented by one double gimbaled CMG from Sperry (model M4500). The reasons for selecting the MACS are two-fold. First, the system provides control within the requirements of the spacecraft. Second, the MACS is a modular system designed to be integrated into different spacecraft and therefore, can be integrated into the LAMR platform.

The CMG spin axis is nominally aligned along the y axis of the spacecraft (Figure 10). This allows the CMG's momentum to be used passively during transfer to damp roll and yaw oscillations. The oscillations in pitch are controlled by pulsing the transfer thrusters which are fired around the center of mass.

Type	3-Axis Stabilized
Attitude Reference	Stellar
Pointing Error	$<10^{-2}$ deg.
Pointing Stability	$<10^{-6}$ deg/sec
Slew Rate (approx.)	1.6 deg/sec
Size (m)	1.2x1.2x0.46
Mass (kg)	215

Table 5 MMS MACS module

Since controlling the spacecraft during transfer is much more difficult than during operations at GEO, the above system has more than sufficient control authority. The orbital transfer ion propulsion system is used to "dump" the momentum build-up in the CMG pitch channel in GEO.

4.4 TELEMETRY, TELECOMMUNICATION AND CONTROL

The Telemetry, Telecommunication and Control (TTC) subsystem uses a modified MMS Control and Data Handling (C&DH) module. The C&DH module has similar characteristics to that used on the Upper Atmospheric Research Satellite (UARS). These properties are shown in Table 6.

Size (m)	1.2x1.2x.46
Mass (kg)	122
Power (Watts)	101
Telemetry Rate	3 Mbps
Command Rate	2 Kbps
Data Storage	1.3x10E9 bit

Table 6 MMS C&DH Abbreviated Specification

4.5 THERMAL CONTROL SYSTEM

The LAMR platform thermal control system is completely passive with the exception of the batteries. The batteries have heaters for the eclipse passage and deep space pointing thermal louvers for the charging period. All of the thermal control on the LAMR is accomplished by the use of thermal insulation and thermal coatings.

The primary reflector, LAMR support boom, feed assembly, and platform bus are covered with Multi-Layer-Insulation (MLI). The use of MLI on the reflector rear, reflector sides, support boom, and feed assembly reduces dimensional instability caused by the rapid and varying heating of the Sun as the platform orbits the Earth. The platform bus is also covered with MLI, producing a thermally-stable environment on the interior of the platform bus.

4.6 STRUCTURES

The LAMR platform has two categories of structures: current and advanced technology. The first category consists of structures used in the construction of the platform bus walls. The second category is comprised of structures beyond the current state-of-the-art, which are required for the instrument.

Current technology structures can be used for many components. The LAMR platform bus (Figure 5) is constructed from a load-bearing shell called monocoque. The shell construction allows for the mounting of equipment on the interior and exterior. The solar array design is taken from the Advanced Solar Array Program (ASAP), designed with the hinged array blanket stretched by a supporting mast (reference 11). The LAMR support boom (Figure 3) is constructed from carbon fiber tubes with aluminum ends connected by aluminum nodes. After the boom is assembled on orbit, the data and power cables are attached to the interior of the box truss, and the boom is covered with MLI. The last major structural element in this category is the feed array assembly (Figure 3). Again, the construction is monocoque shell covered with MLI to provide dimensional stability, structural integrity, and interior volume for the mounting of equipment.

Advanced technology structures are required for the 28-m primary reflector (Figure 3). This reflector is comprised of two elements: the strong back that supports the segmented reflector, and the segments that comprise the reflector surface. Current studies show that an 11-m antenna strong back can be assembled on orbit (reference 7). The scaling of this technology to 28-m may require additional testing. However, the precision reflector segments are the most challenging of the advanced structures. The largest experimental segment that has been produced to date, with the required RMS surface accuracy, is 1-m in diameter (reference 10). Scaling the reflector from 1-m to the required 2-m is difficult. Current experimental segments have spherical reflector surfaces, while the LAMR primary reflector is required to be an offset parabolic surface. To produce this reflector shape, currently used molds will have to be greatly increased in size, which will increase the technological challenge of the 2-m reflector segments. The 4.5-m secondary reflector (Figure 3) is a ground-based spherically-concave tri-segmented reflector. Substantial additional development will be required to transform the current reflector into a parabolic convex reflector.

5.0 ORBITAL TRANSFER

The transfer of the proposed LAMR platform from its assembly point to geostationary orbit is a challenging problem. First, the platform size and mass are larger than previously flown geostationary spacecraft. Such a large payload requires a massive orbital transfer propulsion system which places strong demands on the launch vehicle to place the transfer propulsion system and the platform in LEO. For this reason, minimizing the total mass (platform and orbital transfer system) is the primary design objective.

A second consideration is the acceleration limit imposed by the antenna structural design. The structure is erected in LEO and transferred to GEO in an operational configuration. The mass of antenna structure is greatly reduced by designing it to withstand accelerations limited to less than 0.981 m/s^2 (one-tenth of a g_o).

After two iterations were conducted on system selections and system sizing, an assembled and fueled spacecraft mass of approximately 6,000 kg was defined. This mass is used in the definition of the propulsion system and sizing of the non-platform bus power system. Transfer calculations are based upon a nominal low Earth orbit, 28.5 deg inclination at 500 km, and final orbit of zero degrees inclination, and a geostationary altitude of 35,863 km. One year is chosen as the maximum flight time allowed, such that concerns about the lifetime and reliability of propulsion system components are not deciding factors in choosing the trajectory.

5.1 LOW THRUST CHEMICAL PROPULSION

The large spacecraft mass and the limitation on acceleration requires the use of advanced chemical propulsion systems not presently available. The proposed Orbital Transfer Vehicle (OTV) (reference 12) is designed to fulfill the need for a one-tenth g_o transfer system for large payloads. The 33,566 kg (74,000 lb) optimized space-based OTV is capable of transferring the 6,000 kg platform to GEO. Design specifications for several conceptual systems are also presented in reference 12.

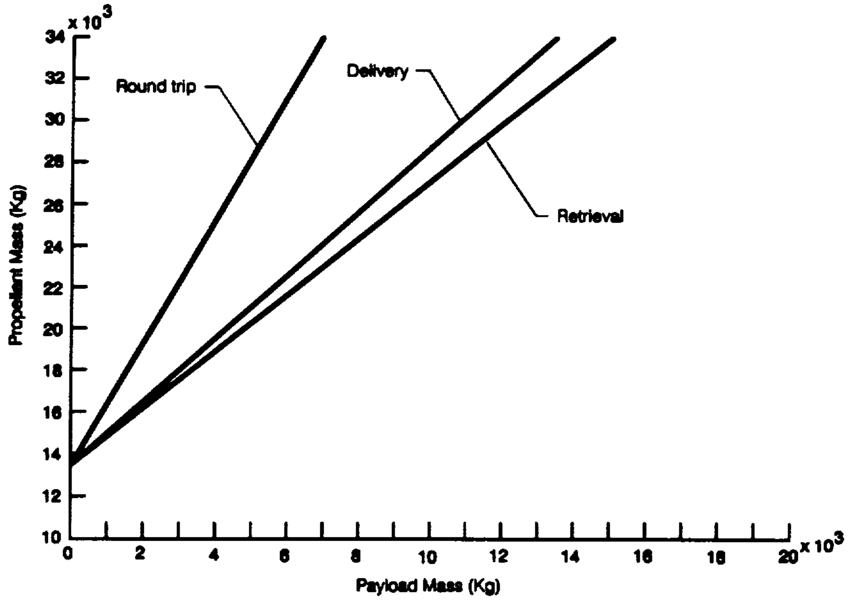

Figure 6 OTV Propellent Mass Estimates Based on Payload Mass

The baseline OTV designs are all reusable spacecraft, so that only the fuel specified for transfer and estimated integration hardware need to be chargeable as LAMR propulsion system launch mass. In this scenario, the OTV and payload are taken from LEO to GEO and the OTV is returned to LEO for reuse. For delivery of the payload and retrieval of the OTV, a total of 21,500 kg of propellent is required to be launched with the LAMR platform components (figure 6). An estimate of 30 percent, or 6,450 kg, is made for the Space Shuttle integration hardware (tanks, payload retaining structure, etc.). Therefore, the total platform and propellent required to be launched to LEO is 33,950 kg (Table 7).

	(kg)
Spacecraft (Payload)	6,000
OTV Propellent	21,500
OTV Propellent Support Hardware	6,450
Total:	33,950

Table 7 Summary of Mass to Orbit for OTV Chemical Propellent Option

5.2 LOW THRUST ELECTRICAL PROPULSION

Three types of low thrust electrical propulsion systems are considered: electric thrusters, nuclear power systems, and photovoltaic power systems.

5.2.1 ELECTRIC THRUSTERS

After consideration of the present technology available for electric propulsion, ion thrusters were chosen for this study. Ion thrusters have a much higher level of technical development compared to magnetohydrodynamic (MHD) engines, and they are more capable of providing continuous thrust than arc-jets or resistojets. Table 8 lists several ion thrusters presently available.

Name	Ref.	Date	Thrust (N)	Power Req. (KW)	Isp (sec)	Lifetime (Hr)
30-cm EMT	(12)	1979	0.13	2.65	3051	15000*
RIT-35	(14)	1972	0.15	4.1	4400	-
ESKA 28	(15)	1972	0.09	2.17	-	-
UK-10(Xe)	(17)	1988	0.07	2.21	4334	-

*predicted

Table 8 Some Representative Ion Thruster Characteristics

Table 9 shows a summary of the proposed thruster, the proposed thruster is representative of Ion thrusters found in current literature. The specifications in this table are used in all of the electrical propulsion trajectory calculations.

Power (P_T)	2 KW
Thrust (T)	0.1 N
Mass* (M_T)	150 kg
P_T/T	20 KW/N
M_T/T	1,500 kg/N
Specific Impulse(Isp)	4,200 sec
Lifetime	8,760 Hr

*Includes the mass of power conditioning equipment and integration hardware.

Table 9 Proposed Electrical Propulsion Thruster

5.2.2 NUCLEAR POWER SOURCES

Nuclear power sources are considered for their ability to provide continuous power over the duration of the transfer trajectory without interruption due to Earth shadowing or power degradation from radiation exposure in the Van Allen belts. Derivatives of the SP-100 program, with a scaled-down output, are considered in this study. An SP-100 derivative with reduced shielding and capable of providing 40 KW_e

of electric power (P_e) is estimated to have a mass ($M_{p/s}$) of 2600 kg (reference 13), which includes the contribution of power conditioning equipment and platform integration hardware. These values give an estimate for the ratio of mass to electric power ($M_{P/S}/P_e$) for an SP-100 derivative of 65 kg/KW_e with a power distribution efficiency (P_T/P_e) of 0.9 and a design lifetime of 10 years.

An estimate of the thrust needed is obtained from Edelbaum's equation (reference 14). Implicit in the use of this equation is that the total platform and propulsion system mass will remain constant. This assumption is valid for the analysis because the ion thruster propellent makes up a small fraction (14% as shown later) of the total system mass in the SP-100 derivative option.

A thrust of 3.6 N (provided by 36 ion thrusters) is required to make the transfer within one year. The spacecraft experiences an acceleration of less than 2 x 10^{-5} g_o on this trajectory. From the specifications given earlier, a power system providing 80 KW_e is required. The complete platform and propulsion system mass at LEO is 19,241 kg, as detailed in Table 10.

A constant mass and no atmospheric drag in LEO were assumed in the trajectory calculations. A constant mass was assumed because of the small portion of total vehicle mass consumed by the propellent. This assumption makes the design conservative since the mass of the spacecraft decreases as the vehicle reaches GEO due to transfer fuel consumption. Atmospheric drag is neglected because at 500 km atmospheric drag is approximately 0.23-N. The atmospheric drag decreases rapidly with increasing altitude. The transfer thrusters have 3.6-N thrust BOL, leaving close to the necessary 3.4-N for the transfer thrust in LEO, then as altitude increases the atmospheric drag will rapidly decrease to zero.

	(kg)
Spacecraft (Payload)	6,000
Thrusters (36)	5,348
Power System (79 KW)	5,135
Propellent	2,758
Total	19,241

Table 10 SP-100 Type Power Option Impact on Spacecraft Mass

5.2.3 PHOTOVOLTAIC POWER SOURCES

In evaluating photovoltaic (PV) sources of power, it is required to estimate the extent to which energetic particles within the Van Allen radiation belts would degrade the power output of these devices, and approximate the effect of the decreasing thrust on the transfer trajectory. A method of estimating this radiation damage makes use of the concept of an equivalent electron flux of 1 MeV (reference 15).

The value of the "equivalent 1 MeV electron flux" at a specific point in space is defined as the flux directed onto a specimen of silicon in the laboratory which imparts the same damage onto that specimen as would be imparted by the full spectrum of energetic particles at the specific point in space. A shield thickness of 12 mil was used for relief from radiation degradation of the photovoltaic power system when it was exposed to high levels of radiation for long duration (equivalent to the trip through the Van Allen belts).

This theory has only been fully developed for silicon PV cells. Also, only the damage resulting from exposure of the front surface of the arrays is included in the current analysis. The change in cell efficiency with operating temperature is also neglected. Therefore, it must be remembered that the results obtained in this analysis are only for a comparison of the effects of exposure to high-energy particles between several options for the platform propulsion system.

The decrease in available PV electrical output power $P_e(T)$ is given as a function of time. This function is obtained from the fractional loss of power (P_e/P_o), which is an empirically determined function of the cumulative 1 MeV fluence received over the trajectory up to the time T. Figure 7 shows the function $P_e/P_o(F)$ for conventional 2 ohm-cm n/p silicon cells (reference 15) and for advanced indium phosphide (InP) cells (reference 16). Figure 7 shows that indium phosphide has less power loss at higher exposure levels than silicon.

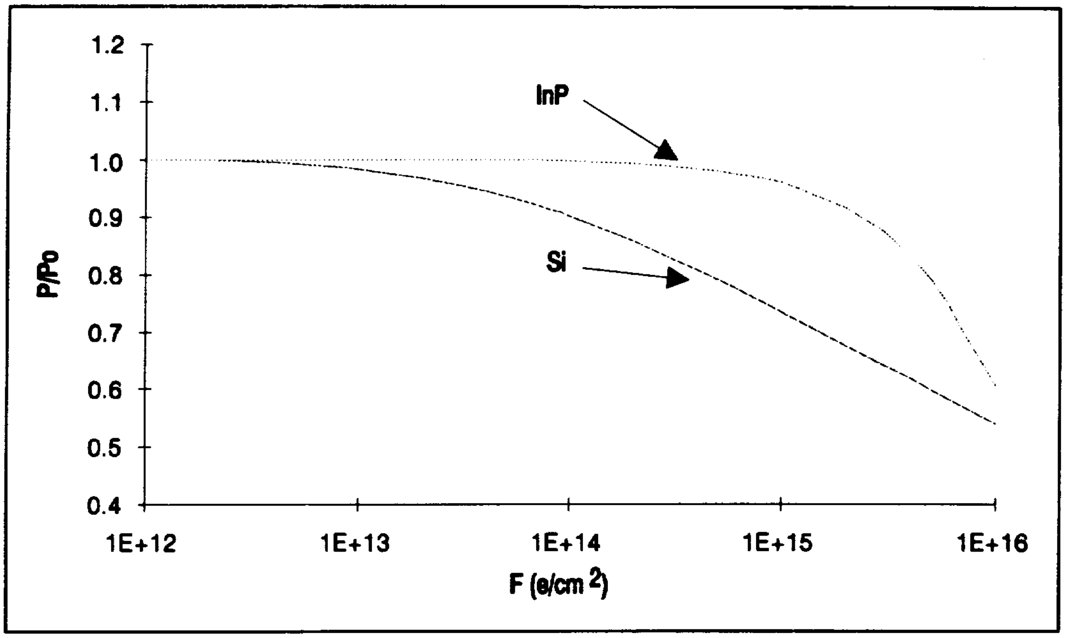

Figure 7 PV Electrical Power Loss versus 1 MeV Fluence

Trajectories are integrated using the Program to Optimize Simulated Trajectories (POST) (reference 17). The analysis is performed to reach GEO in one year without optimization. The computer model assumes a loss of power to the thrusters when the spacecraft enters the Earth's shadow. This eliminates the requirements for onboard power storage for transfer propulsion purposes.

Two trajectories are analyzed, one for the platform utilizing the conventional 2 ohm-cm n/p silicon cells (reference 15) and the other for the platform utilizing the newer InP technology (reference 16). Ion thrusters, with the same performance as those selected in section 5.2.1, are used. The impact on the mass of the integrated platform is summarized in Table 11. The results for each trajectory are summarized in Table 12.

	Silicon	**InP**
Payload mass (kg)	6,000	6,000
Thruster mass (kg)	12,488	5,778
Power system mass (kg)	1,542	713
Propellent Mass (kg)	4,784	2,213
Total spacecraft mass	24,814	14,704

Table 11 Solar-Electric Trajectories Mass Results

	Silicon	InP
Cell type	Si 2 ohm-cm	InP- n/p
Shield thickness (mil)	12	12
Initial power (KW)	185	86
Initial thrust (N)	8.3	3.8
Final power (KW)	105	62
Final thrust (N)	4.7	2.8
Power system (kg/KW)	120	120
Power system efficiency	0.9	0.9
Number of thrusters	83	38
Transfer time (days)	366	363
Total fluence (e/cm^2)	7.8X10E15	6.6X10E5

Table 12 Solar-Electric Trajectories Performance Characteristics

Figure 8 shows the cumulative electron flux received by the power system as a function of time, and Figure 9 shows the corresponding decrease in thrust.

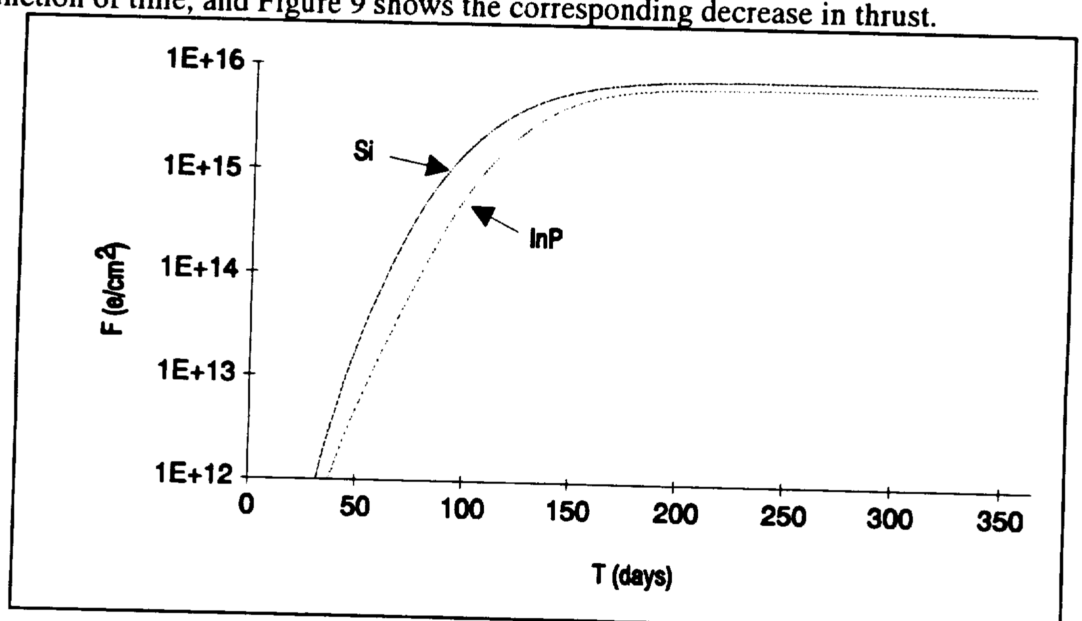

Figure 8 Cumulative Fluence versus Orbital Transfer Time

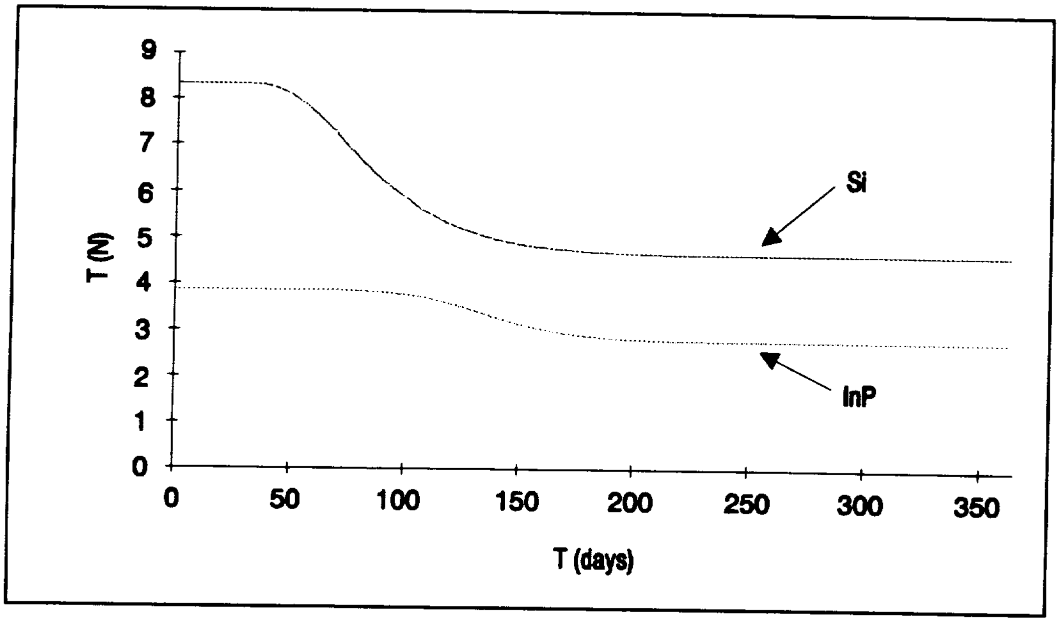

Figure 9 Thrust versus Orbital Transfer Time

As shown in Table 9, a solar electric propelled transfer vehicle with payload at LEO has a mass of 24,814 kg if it utilizes conventional silicon cells, or a mass of 14,704 kg if the advanced indium phosphide cells are employed. Since the indium phosphide electrical ion propulsion option meets all the platform requirements and has the least mass of the four options considered (low thrust chemical, nuclear electric, silicon electric and indium phosphide electric), it is proposed for the conceptual design of the LAMR platform and its systems.

6.0 PLATFORM ATTITUDE CONTROLS ANALYSIS RESULTS

The LAMR platform has two distinct operational modes. Each mode has special challenges with respect to attitude control. After the LAMR components are delivered to orbit, assembled, checked out and released, they begin their year-long journey to geostationary orbit. This period of orbital transfer is the first operational mode. When the LAMR platform is at GEO, the platform positions the sensor axis towards the Earth and begins data collection, which is the second operational mode. Both modes are described with respect to a body fixed coordinate system, shown in Figure 10, where $\mathbf{X_0}$, $\mathbf{Y_0}$, and $\mathbf{Z_0}$ are inertially fixed with an origin at the center of the Earth.

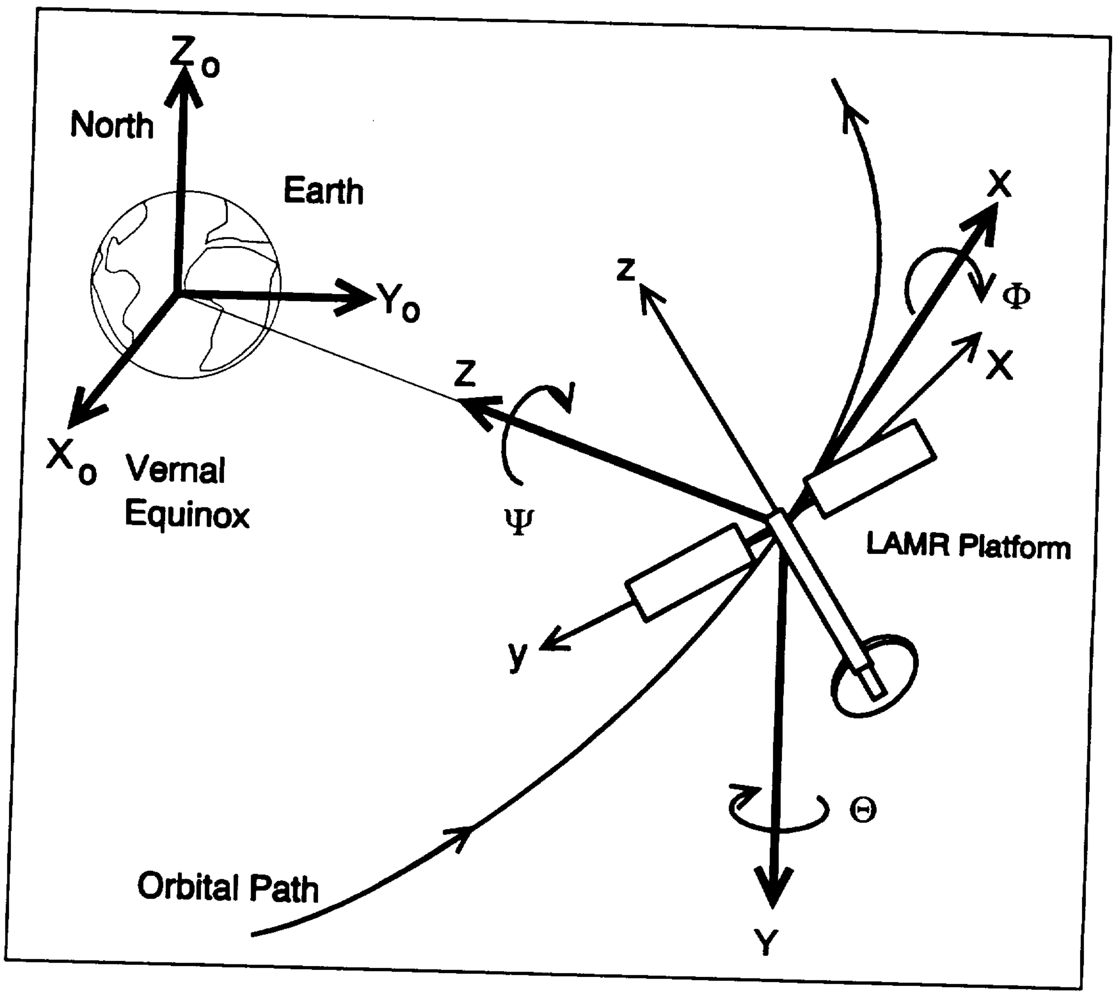

Figure 10 Definition of a Body Fixed Coordinate System

The orbital coordinate system **X**, **Y** and **Z** has its origin at the platform center of mass and describes the orbital motion of the platform. The **X** axis is always along the velocity vector. The **Z** axis points toward the center of the Earth. And, the **Y** axis is

perpendicular to the orbital plane. The body-fixed coordinate system is represented by the axes **x**, **y** and **z** on the platform. Here, the **z** axis is aligned along the sensor or primary viewing axis (section 3.0). The y axis is aligned along the solar array axis. And, the **x** axis is mutually perpendicular to the other two axes. Three primary rotations, Ψ, Θ, and Φ, orient the body fixed coordinate system with respect to the orbital coordinate system. Ψ, Θ and Φ are rotations of the orbital coordinate system about **Z**, **Y** and **X**, respectively, and are called Euler angles.

The flight attitude of the platform in LEO during orbital transfer is shown in Figure 11.

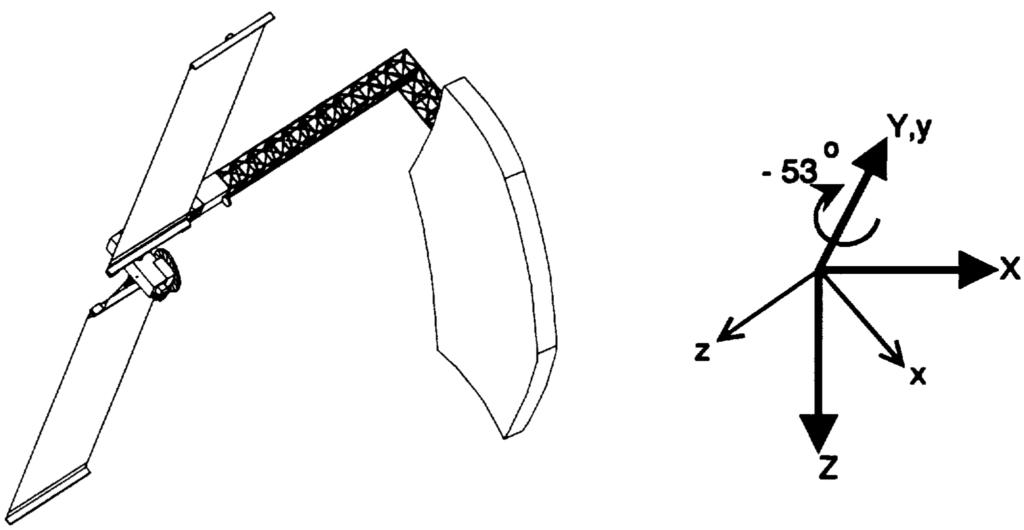

Figure 11 Low Earth Orbit Control Attitude

In this flight mode an initial rotation of -53 degrees about the **Y** axis is performed. This orientation places the center of pressure ahead of the center of mass, resulting in an unstable configuration. The preferred flight attitude in LEO is with the center of pressure trailing the center of mass. This configuration is stable in yaw, but any orbital debris will impact on the primary reflector surface. Therefore, the alternate attitude, with the deflector face opposite the velocity vector, is chosen. This attitude is shown in Figure 11. The attitude is derived from the balance of environmental torques on the platform, this balanced attitude is refereed to as a Torque Equilibrium Attitude (TEA). The initial Euler angle (Θ=-53 degrees) allows the spacecraft to balance the gravity gradient torque against the aerodynamic torque. This results in a control torque that is acceptable for the control system selected. Results of the low earth operational attitude mode are shown in Figures 12 and 13.

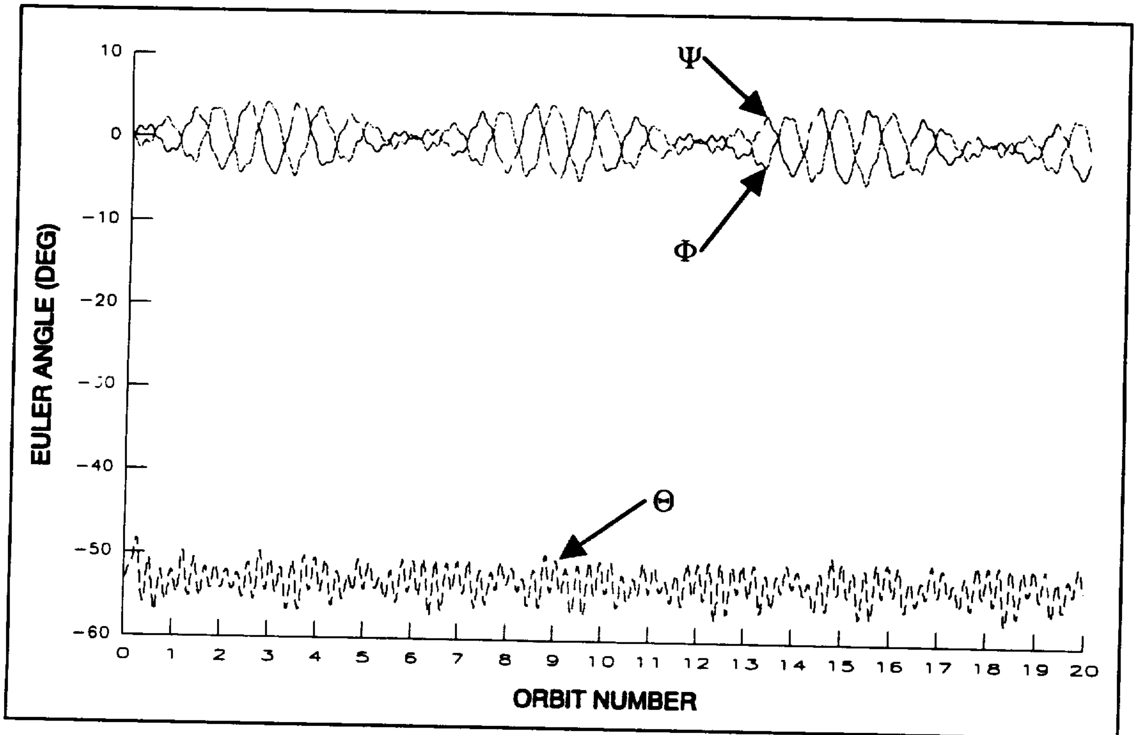

Figure 12 Euler Angles versus Orbit Number (LEO)

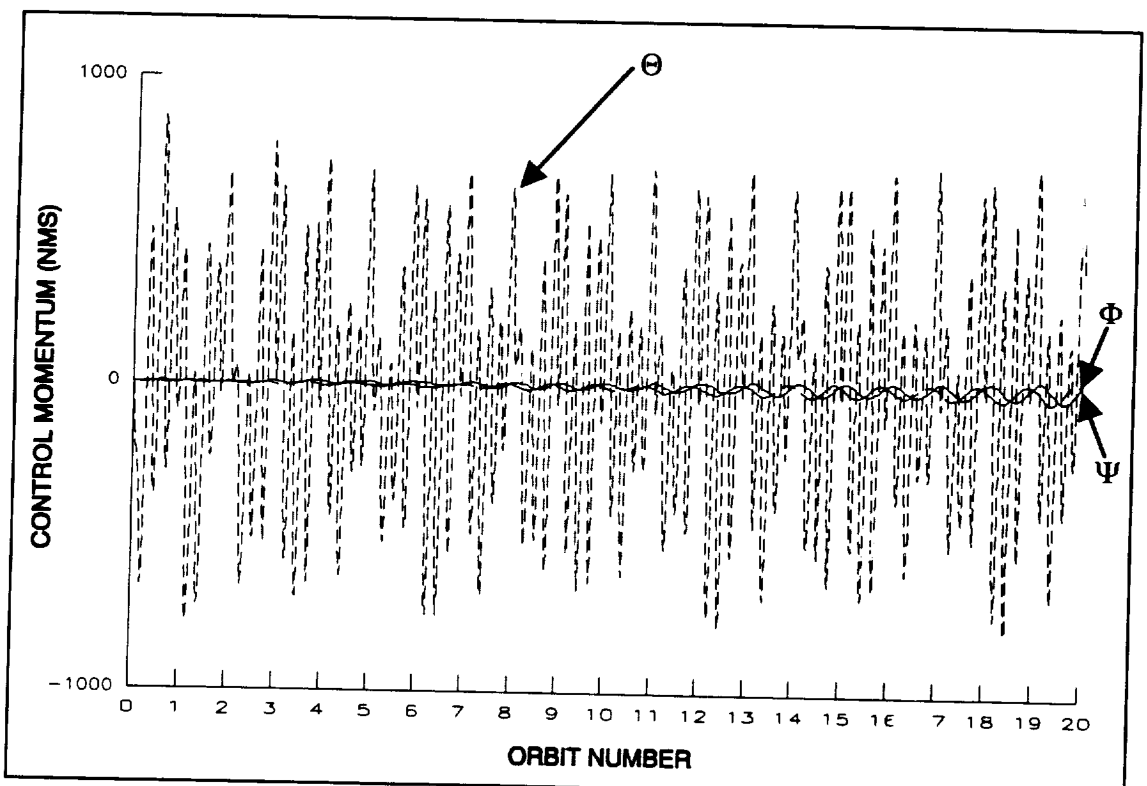

Figure 13 CMG Angular Momentum versus Orbit Number (LEO)

Figure 12 shows that the Euler angles stay within a 15 degree dead band over a 20-orbit period. In order to control this vehicle with the available CMG control laws, the vehicle was modeled with a momentum exchange device, with an angular momentum capacity of 6100 Nms, aligned along the pitch axis (**y** axis, Figure 10) and a flywheel, of the same capacity, aligned for roll and yaw (**x** and **z** axis respectively, Figure 10). The pitch channel was decoupled from the roll and yaw channels so that the CMG does not affect the roll/yaw. Since there is in fact a small coupling between pitch and roll/yaw, the use of a control law specifically designed for this platform's unique flight attitude will more accurately describe its motion. Figure 13 shows the CMG angular momentum over the same 20-orbit period. The momentum stays within a 1800-Nms dead band with a peak requirement of 900 Nms, which is within the design limits of the selected CMG.

The next altitude examined for the platform is in the Van Allen Belts. At this altitude, 3189 km, there is a limited atmosphere and therefore, minimal atmospheric drag to balance the gravity gradient torques. Although the gravity gradient forces are dominant, their magnitude decreases substantially from LEO. Also, there are small contributions from the solar radiation forces. The platform TEA, at this altitude requires an initial Euler angle of Θ=-61.5 degrees. Figure 14 shows virtually flat Euler angles for a 10-orbit period.

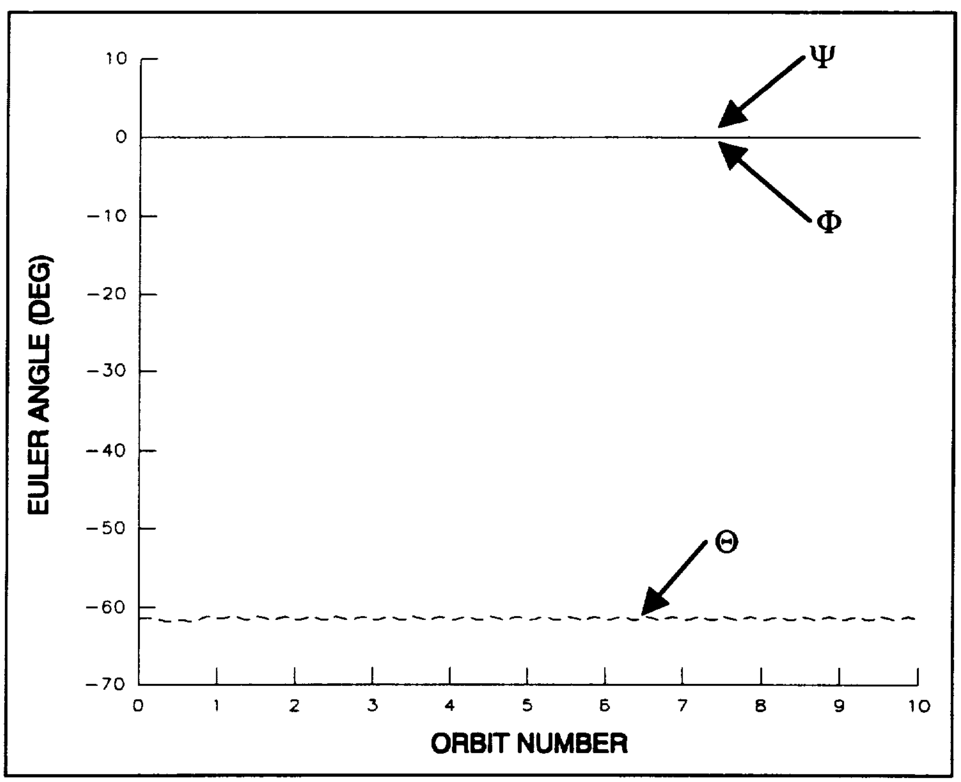

Figure 14 Euler Angles versus Orbit Number (3189 km Altitude)

This results from the platform being in a relatively benign attitude and the environmental forces at this altitude being substantially smaller than those in LEO. Φ and Ψ are approximately equal to zero. Therefore, the control system can easily control the platform at this altitude and attitude.

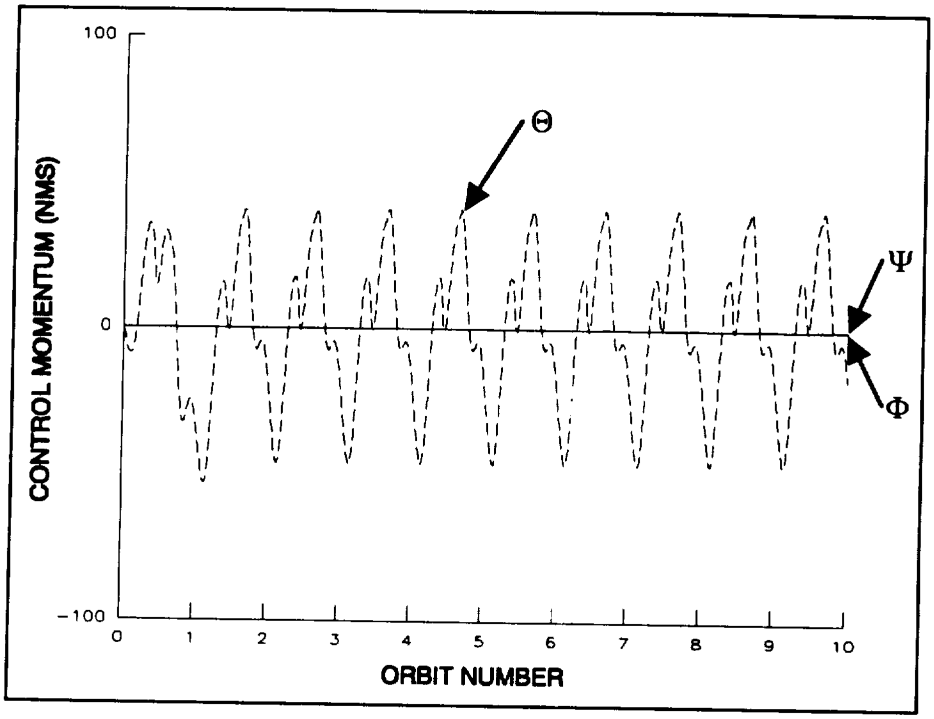

Figure 15 **CMG Angular Momentum versus Orbit Number (3189 km Altitude)**

Figure 15 shows that the CMG angular momentum operates within a dead band of approximately 80 Nms, well within the design limits of the chosen CMG. After an initial period, Θ settles into a regular periodic oscillation. Φ and Ψ are approximately equal to zero due to no out-of-plane torques.

At GEO, the platform instruments will commence operations. In the operational attitude the Euler angles are all zero (to align the sensor axis to Nadir, Figure 16).

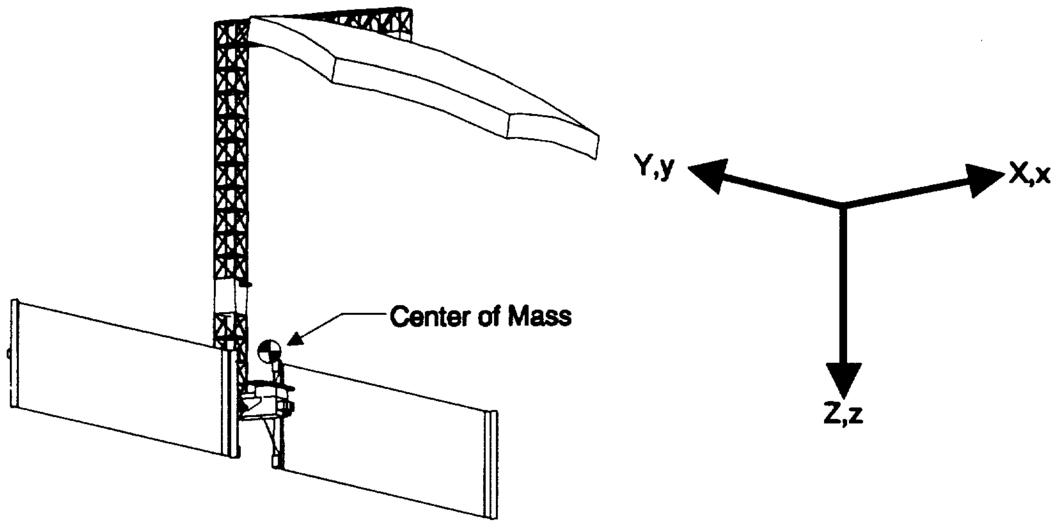

Figure 16 LAMR Platform Attitude at GEO

The resulting environmental forces (predominantly solar radiation forces) yield a secular momentum build-up, which must be stored by the CMG and later dissipated by the RCS system before the CMG saturates. The attitude of the platform at GEO is not near a TEA, therefore a net angular momentum build-up can not be avoided. The control requirements are shown in Figures 17 and 18.

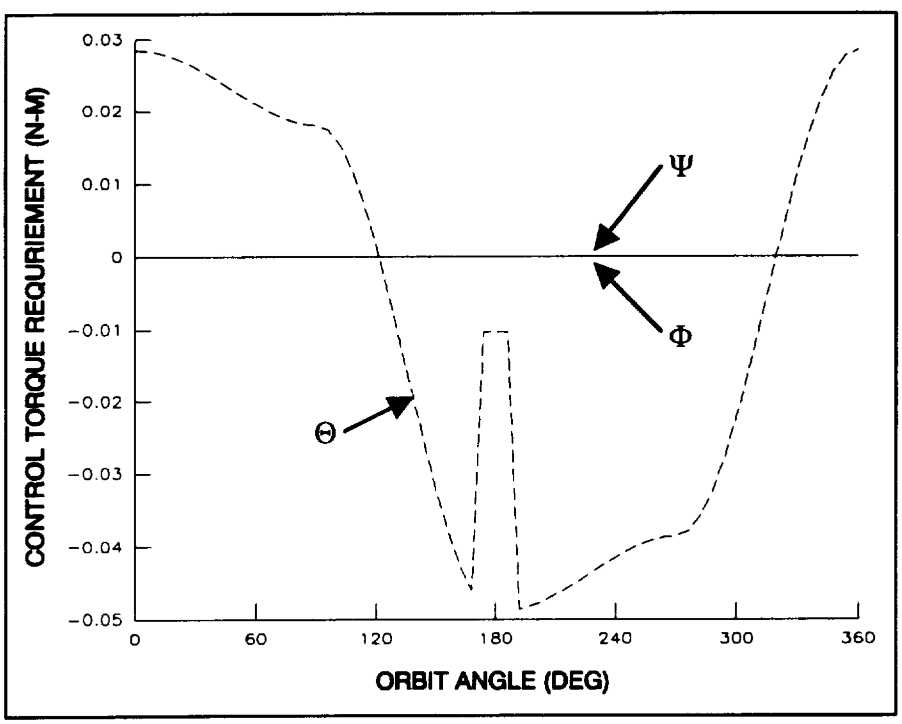

Figure 17 Control Torque Requirements versus Orbit Angle (GEO)

Figure 17 shows the platform control torque requirement over a single orbit period. The pitch control torque requirement is illustrated in Figure 17, where the discontinuity in the control torque results from the platform entering the Earth's shadow. The pitch control torque does not got to zero because of the gravity gradient torque.

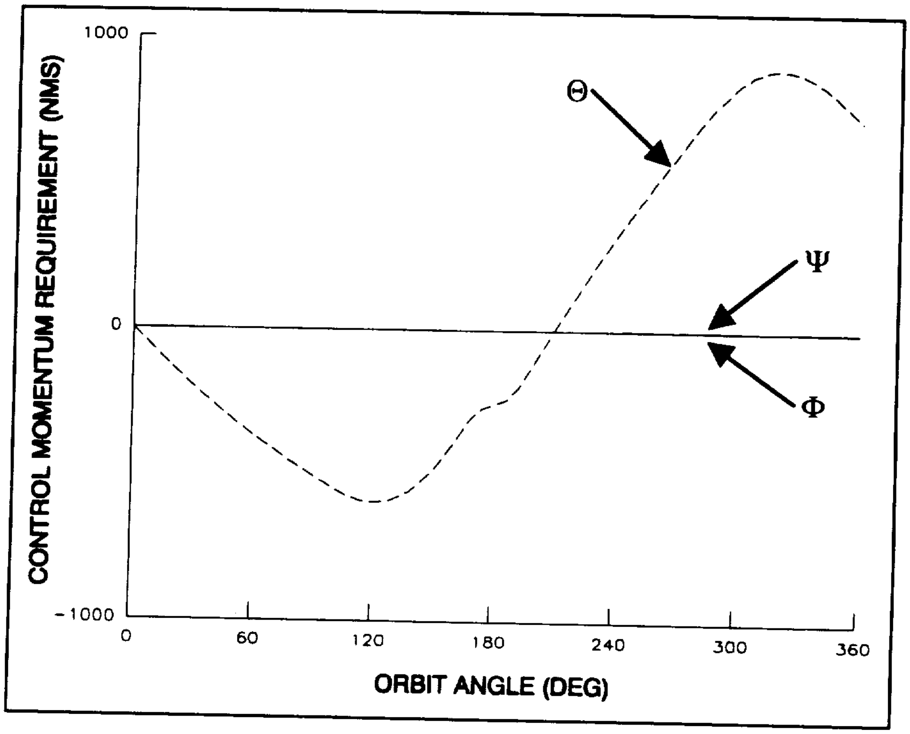

Figure 18 Control Momentum Requirements versus Orbit Angle (GEO)

Figure 18 shows the control momentum requirement over a single orbital period. There is a large angular momentum build-up around the pitch axis. This secular build-up is due to the orientation of the principal axis relative the flight mode and the solar radiation momentum. In order to remove this build-up, some momentum management must be performed every orbit. The ion propulsion system is used to produce a negative pitch torque to counteract the positive pitch build-up. The thrusters on the -x side of the CG are fired for approximately an hour each orbit to reduce the momentum build-up prior to CMG saturation.

7.0 ENABLING TECHNOLOGIES

The conceptual design for a LAMR platform presented in the previous sections uses many advanced technologies in its design. These enabling technologies and technology areas are detailed below.

In the area of solar array technology, two aspects must be improved upon to enable this and many other spacecraft to complete their missions: solar radiation degradation and the mass of solar arrays. As previously described, radiation degradation of solar cells presents a large impediment to solar electric low thrust transfer from LEO to GEO. Silicon, although used extensively for spacecraft, does not provide enough protection against this radiation degradation. The most promising solar cell type is the indium phosphide. The use of indium phosphide would reduce the surface area and the mass of the solar arrays by as much as 50 percent. It has an efficiency of 11 percent, with a specific power range of between 130 to 180 W/kg. The resistance to solar radiation power loss for the indium phosphide was found to be three times that of silicon for a sample 200-day LE0-to-GEO spiral orbit transfer mission (reference 7). This translates into smaller arrays, less power system mass, and less propulsion system mass due to the overall decrease in the platform mass.

Integrated erectable or deployable space structures are another area requiring improvement. Although the basis for the LAMR platform design is an erectable structure, no ground or flight experiment program has demonstrated the assembly and checkout of a complex system in a simulated or true space environment to date. Here, the reference to complex structures is to include systems like instrument wiring harnesses, power cable runs, fluid transport mechanisms, and spacecraft subsystem components. Previously, studies have not integrated these complex systems. The studies have, however, defined the assembly of reflector surfaces and support structures which comprise the skeleton of space systems, but have neglected the interior and exterior equipment. This is the area in which significant technology development must be achieved.

An analogy to this situation is the construction of a house. The amount of time and effort spent framing a house is very small as compared to the laying of electrical runs, plumbing, roofing, plastering, siding and final trimming and painting. The challenge is to design and produce integrated systems that astronauts and robots can assemble on-orbit. These complex systems constitute an entire technology area that will require ground testing and on-orbit simulation.

Antenna reflector technology are the most critical area that needs improvement. Large antennas with high-frequency requirements such as the LAMR will require high surface accuracy and solid reflectors. There are three technology concerns in this area: size, shape, and areal density (the mass of an antenna divided by its reflector surface area). The LAMR primary reflector is 28 m in diameter. To reduce the number of segments that comprise the surface and also span the distance between nodes of the support structure, segments will have to be on the order of 2 m in size. Current programs are working with spherical surfaced panels ranging from 0.3 to 1.0 m across. These panels have the required surface accuracy, but they are too small to fulfill the required nodal span. Also, the shape of the surface required is offset parabolic. Only spherical panels have been produced to date. Researchers estimate development of 2-m offset paraboloid panels is at least an order of magnitude more difficult than current panel development.

Other systems on the LAMR could accommodate systems that do not have the technological advancement requirements set forth here. An example of this could be power generation systems that are less efficient in the envisioned environment, thus being much larger and heavier. These single system impacts would drive the costs up substantially (in terms of weight to orbit and time to operational station) . But, the need for panels in the 2-m range is essential to the viability of this mission.

The areal densities of large space antennas are currently on the order of 5 kg/m^2 which is the state-of-art for large space antennas. The LAMR has a reflector with approximately 616 m^2 of surface area, which translates into the primary reflector mass of 3,100 kg. The primary reflector becomes nearly half the mass of the platform. The increase in mass has a ripple effect of increasing the mass of each system. The larger mass of the primary reflector has a destabilizing effect on the attitude control system. The larger mass of the primary reflector (30 m from the spacecraft bus) will decrease the effects of the gravity gradient torques which are used to dissipate the secular momentum build-up due to solar radiation pressure. It also increases the requirements of the attitude control system with respect to accommodation of the cyclic momentum build-up of the LAMR platform. The dramatic platform mass increase from higher areal densities will translate into an explosive mass and system requirement for the propulsion transfer system. Therefore, the reflector panel surface, size and density are critical technology elements of the LAMR platform.

8.0 CONCLUDING REMARKS

The major purpose of this study was to point to technologies and technology areas that are essential to the design and on orbit viability of the LAMR platform. The technologies include advanced solar cells, integrated complex structures, large segmented reflector panels, lower antenna areal densities, and ion propulsion. Each of these technologies must be developed to the requirements in this paper for the LAMR mission to be viable.

The LAMR platform will require advanced solar cells that can maintain power when exposed to high levels of radiation and retain high specific power. Current technologies include silicon and gallium arsenide base cells. Although they have high BOL specific power, they will not withstand the radiation flux required by the LAMR platform. Indium phosphide based solar cells will meet both criteria, but at this point are not fully developed.

The sensor and platform will involve assembly of thousands of parts into an integrated complex structure. Currently, only skeletons of single reflectors are under study. Integrated complex structures technology will require substantial development and ground testing.

The segmented solid reflector panels must span 2 m, have offset parabolic shapes and still maintain RMS surface accuracy of 1×10^{-4} m. Current panels of 1 m or less and spherical in shape are being tested. Panel technology will have to surmount these hurdles for the LAMR platform to be viable.

The LAMR platform will require an antenna areal density of 3 kg/m^2 or less with the above panel requirements. Advanced antennas today have an areal density of 5 kg/m^2. Current areal densities would drive overall platform mass past viable mass limits.

The propulsion requirements of the LAMR platform dictate a massive system with continuous engine lifetimes of years. Current ion propulsion technology has been demonstrated in ground testing. However, these systems are much smaller than the one necessary and the lifetimes demonstrated are one year or less.

Technology areas that require further development are electrical power storage, on-orbit assembly, and on-orbit systems checkout and correction. These would enhance the capabilities of the LAMR platform, but advancements are not essential for viability.

9.0 REFERENCES

1. Pioneering the Space Frontier, *The Report of the National Commission on Space*, Bantam Books, 1986.

2. S. K. Ride, "Leadership and America's Future in Space - A Report to the Administrator", August 1987.

3. W. Stutzman and G. Brown, "The Science Benefits of and the Antenna Requirements for Microwave Remote Sensing From Geostationary Orbit", NASA CR-4408, October 1991.

4. C. Rogers and W. Stutzman, "Large Deployable Antenna Program *Phase I: Technology Assessment and Mission Architecture*", NASA CR-4410, October 1991.

5. W. Stutzman and G. Thiele, *Antenna Theory and Design*, John Wiley:New York, 1981.

6. Thomson, Mark, Deployable Reflector Development for LDA, PR-324, Astro Aerospace Corporation, Carpinteria, CA, 22 February 1991.

7. H. G. Bush, et al, "Design and Fabrication of an Erectable Truss for Precision Segmented Reflector Application", *Journal of Spacecraft and Rockets*, Vol. 28, p 251-257, March-April 1991.

8. F. Grimaldi, et al, "Development of a Large Deployable Carbon Fiber Composite Antenna Structure for Future Advanced Communications Satellites", *Materials and Processing - Move into the 90's*, Edited by S. Benson, et al, Elsevier Science Publishers B.V., Amsterdam, 1989.

9. B. Agrawal, *Design of Geosynchronous Spacecraft*, Prentice-Hall, Englewood Cliffs, NJ, 1986.

10. R. Freeland, "Deployable Space Antenna Technology State of the Art", Telescope Optical Systems Meeting, NASA LaRC, Hampton, VA , 25 June 1991.

11. P. Stella and R. Kurland, "Latest Developments in the Advanced Photovoltaic Solar Array Program", Proc. 25th Intersociety Energy Conversion Engineering Conference, Vol. 1, pp. 569-574, August 1990.

12. A.E. Kofal, "Orbital Transfer Vehicle Concept Definition and System Analysis Study, Volume IX Study Extension Results", NASA CR-183549, 1986.

13. M.S. El-Genk, M.D. Hoover, "A summary Overview of Recent Advances in Space Nuclear Power Systems Technology", Proc. 25th Intersociety Energy Conversion Engineering Conference, Vol 1, pp. 100-105, August 1990.

14. T.N. Edelbaum, "Propulsion Requirements for Controllable Satellites", *American Rocket Society Journal*, pp. 1079-1085, August 1961.

15. H.Y. Tada, et al., *Solar Cell Radiation Handbook*, JPL Publication 82-69, November 1, 1982.

16. P.M. Stella, D.J. Flood, "Photovoltaic Options for Solar Electric Propulsion", NASA TM-103284, 1990.

17. G.L. Brauer, et al., "Program To Optimize Simulated Trajectories (POST)", Vol. II, Martin Marietta Corporation, NAS1-18147, September 19, 1989.

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY *(Leave blank)*

2. REPORT DATE
June 1992

3. REPORT TYPE AND DATES COVERED
Technical Memorandum

4. TITLE AND SUBTITLE
A Conceptual Design of a Large Aperture Microwave Radiometer Geostationary Platform

5. FUNDING NUMBERS
506-49-21-02

6. AUTHOR(S)
Paul A. Garn, James L. Garrison, and Rachel Jasinski

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
NASA Langley Research Center
Hampton, Virginia 23665-5225

8. PERFORMING ORGANIZATION REPORT NUMBER

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Washington, DC 20546

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
NASA TM-107577

11. SUPPLEMENTARY NOTES
P. A. Garn, Science & Technology Corp., Hampton, VA; J. L. Garrison, Langley Research Center, Hampton, VA; R. Jasinski, Flight Mechanics & Control, Corp., Hampton, VA

12a. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified - Unlimited

Subject Category 18

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

A conceptual design of a Large Aperture Microwave Radiometer (LAMR) Platform has been developed and technology areas essential to the design and on-orbit viability of the platform have been defined. Those technologies that must be developed to the requirement in this paper for the LAMR mission to be viable include: advanced radiation resistant solar cells, integrated complex structures, large segmented reflector panels, sub 3 kg/m^2 areal density large antennas, and electric propulsion systems. Technology areas that require further development to enhance the capabilities of the LAMR platform (but are not essential for viability) include: electrical power storage, on-orbit assembly and on-orbit systems checkout and correction.

14. SUBJECT TERMS
Spacecraft design, Antennas, Microwave, Radiometer, Remote sensing

15. NUMBER OF PAGES
37

16. PRICE CODE
A03

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	UL